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# An Analytic Approximation to Very High Specific Impulse and Specific Power Interplanetary Space Mission Analysis

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# **AN ANALYTIC APPROXIMATION TO VERY HIGH SPECIFIC IMPULSE & SPECIFIC POWER INTERPLANETARY SPACE MISSION ANALYSIS**

**Craig Hamilton Williams\***

A simple, analytic approximation is derived to calculate trip time and performance for propulsion systems of very high specific impulse (50,000 to 200,000 seconds) and very high specific power (10 to 1000 kW/kg) for human interplanetary space missions. The approach assumed field-free space, constant thrust/constant specific power, and near straight line (radial) trajectories between the planets. Closed form, one dimensional equations of motion for two-burn rendezvous and four-burn round trip missions are derived as a function of specific impulse, specific power, and propellant mass ratio. The equations are coupled to an optimizing parameter that maximizes performance and minimizes trip time. Data generated for hypothetical one-way and round trip human missions to Jupiter were found to be within 1% and 6% accuracy of integrated solutions respectively; verifying that for these systems, credible analysis does not require computationally intensive numerical techniques.

## **INTRODUCTION**

Through countless millennia, mankind has ventured to unknown worlds in the pursuit of wealth, knowledge, freedom, adventure, and international prestige. To date, human exploration of space has been primarily driven by the pursuit of international prestige. Because of the high cost and long trip times associated with human interplanetary exploration, such endeavors are expected to remain unattractive to any potential sponsor other than the governments of the most financially capable and technologically advanced nations. While it is difficult to imagine a low cost solution to human interplanetary travel, far-term technically feasible concepts do exist that could significantly reduce the expected long trip times. This could obviate larger and more complex spacecraft, higher operations costs, inordinately long return on investment times compared to commercial ventures, and the unappealing sacrifice of the spacefarer's personal time. Conventional chemical, and nearer-term electric and nuclear fission propulsion technologies would require multi-year round-trip times to even the closest planets. Although long duration missions have been the norm for unmanned probes, they represent a real hardship for human travel. More advanced propulsion technologies, however, could provide trip times comparable to some commercial terrestrial operations --- several weeks, perhaps even days. As D. Cole pointed out in his 1959 paper on minimum trip time travel to the planets, the evolution of terrestrial propulsion systems has been from the inexpensive "minimum energy" sailing vessels to the more costly "minimum time" aircraft<sup>1</sup>. Advanced space propulsion systems could catalyze future interplanetary travel predicated on the private, not public, sector market. High specific impulse ( $I_p$ )/high specific power ( $\alpha$ ) systems could accomplish this by traversing almost straight line trajectories between planets due to their expected ability to generate large accelerations. These propulsive technologies, unfortunately, are well beyond near-term feasibility and require solving some formidable technical challenges.

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Because of the technological immaturity and perceived remoteness of these advanced propulsion concepts, most human interplanetary mission studies have been predicated on trip time and propulsion system performance data generated by analysis tools designed to model either  $I_{sp}$ -limited systems (high thrust) or  $\alpha$ -limited systems (low thrust). As a result, high  $I_{sp}$ /high  $\alpha$  technologies are infrequently studied and therefore have few analytic tools designed to model them. Fortunately, due to the nature of their operation, simple analytical techniques can provide reasonably accurate estimates of their mission travel times and performance capabilities. These simple, first-order analytical methods can greatly assist in focussing development of requisite precursor technologies. Very high thrust propulsion systems could produce accelerations greater than the local acceleration due to solar gravity at Earth's orbit ( $0.6 \text{ E-3 g}$ ). The normally thought-of conics of minimum energy trajectories degenerate into straight line transfers at these acceleration levels. A "field-free space" approximation can then be invoked to greatly simplify the usually complex orbital mechanics. Gravity losses and optimum steering concerns can be neglected without introducing too much error, obviating the need for computationally intensive, numerically integrated solutions to support a proof of concept analysis. As a result, an analytic closed-form solution can be derived that calculates minimized trip time and maximized performance capability in a single equation (or at most two).

An extensive literature search surfaced three published approaches that solve only select portions of this problem for a constant thrust engine; though it is also worth noting that a few papers have also been written on the simpler to analyze, constant acceleration (variable thrust) device<sup>1</sup>. During the 1950's, H. Preston-Thomas of the National Research Council of Canada derived several one dimensional velocity equations<sup>2</sup>. Found just prior to the publication of this paper, Preston-Thomas' work contains trip time relations, but sparse derivations and few equations of motion made assessment of the approach difficult. Wolfgang Moeckel of the NASA Lewis Research Center developed a simple analytic approach using a one dimensional "flat solar system" model to calculate trip time using the quotient of velocity and distance during the 1960's<sup>3</sup>. High and low thrust systems were treated separately; the high thrust relations based on vehicle  $\Delta V$  calculated by the classic rocket equation, the low thrust relations based on the time integral of acceleration squared equation ("J" parameter), modified to optimize engine-on time. Moeckel's approach had the advantage of extreme simplicity with reasonably good accuracy. However, the segregation of high and low thrust systems into separate equations did not illustrate the explicit dependence of trip time as a function of only distance,  $I_{sp}$ , and  $\alpha$  in advanced very high  $I_{sp}$ /very high  $\alpha$  systems.

Professor Dennis Shepherd of Cornell University developed an alternate approach in the 1960's<sup>4</sup>. Based in part on some earlier work by Ernst Stuhlinger, Shepherd's approach used a one dimensional, finite burn model (continuous thrusting with no coast phases). He derived a closed-form equation for distance traveled in field-free space by an accelerating rocket at constant thrust in terms of  $I_{sp}$ ,  $\alpha$ , and trip time. The classic rocket equation served as the starting point, later incorporating an optimizing parameter to maximize performance and minimize trip time. One advantage of Shepherd's finite burn approach was a more conservative representation of one dimensional motion. More important, however, was the explicit appearance of  $I_{sp}$  with  $\alpha$  in a single equation optimized for payload and trip time, thus providing the analyst with a convenient way of evaluating the relative effect of distance,  $I_{sp}$ , and  $\alpha$  on trip time for any mission distance. The main limitation was that the relation was valid for one-way flyby missions only, where the spacecraft would not rendezvous with the destination planet.

The problem solved in this paper is an extension of Shepherd's work. Two-burn "rendezvous" and four-burn "round trip" mission equations of motion are derived with an optimization condition imposed. Each leg of the trip consists of an acceleration burn from Earth's heliocentric position followed immediately by a deceleration burn (with no intervening coast period) which terminates at the heliocentric position of the destination planet (with zero radial velocity with respect to the sun). The transfers are assumed to originate and terminate outside the effective planetary gravity wells. These one dimensional straight line approximations along the heliocentric radius vector closely resemble numerically integrated solutions (Figure 1).

## APPROACH

### Review and Reformulation of the Solution to the One-Burn Flyby Problem

Shepherd's <sup>4</sup> approach will be briefly summarized. The basic rocket equation for a constant  $I_{sp}$  vehicle is:

$$\Delta V = c \ln \left( 1 - \frac{M_p}{M_i} \right)^{-1} = c \ln(1 - \lambda)^{-1} \quad (1)$$

The mass balance relation is given by:

$$M_p + M_s = M_i - M_{pay} \quad (2)$$

Rearranging terms of the mass equation yields:

$$M_p \left( 1 + \frac{M_s}{M_p} \right) = M_i \left( 1 - \frac{M_{pay}}{M_i} \right) \quad (3)$$

Substituting for the propellant mass ratio into the rocket equation yields:

$$\Delta V = c \ln \left[ \frac{1 + \frac{M_s}{M_p}}{\frac{\frac{M_{pay}}{M_i} + \frac{M_s}{M_p}}{1}} \right] \quad (4)$$

From the definitions of jet power and specific power, a structure to propellant mass ratio can be defined:

$$P = \frac{\dot{M}_p c^2}{2g_c} \quad (5)$$

$$\alpha = \frac{P}{M_s \eta} \quad (6)$$

$$\text{thus: } \frac{M_s}{M_p} = \frac{c^2}{2g_c \eta \alpha T} \quad (7)$$

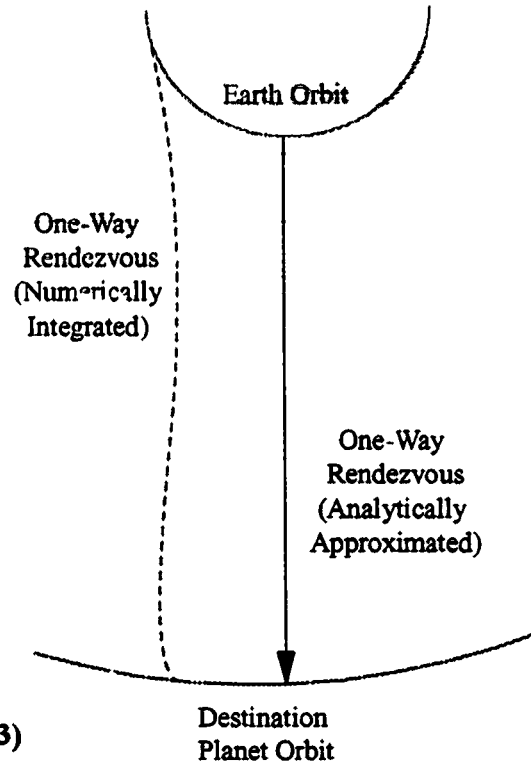


Figure 1: Continuously Thrusting Trajectories

Shepherd introduced a parameter he called the *characteristic velocity*  $V_c$ , which upon substitution allows for a useful expression for the structure to propellant mass ratio:

$$V_c = \sqrt{2g_c \eta \alpha T} \quad (8)$$

$$\text{thus: } \frac{M_s}{M_p} = \left( \frac{c}{V_c} \right)^2 \quad (9)$$

Substituting Eq. (9) into Eq. (4) yields a relation for payload ratio in terms of  $\Delta V$ , exhaust and characteristic velocities:

$$\frac{M_{pay}}{M_i} = \frac{1 + \left( \frac{c}{V_c} \right)^2}{\exp \frac{\Delta V}{c}} - \left( \frac{c}{V_c} \right)^2 \quad (10)$$

Shepherd illustrated through a series of plots the implications of such an expression, primarily the existence of an optimal  $I_{sp}$  (in which the payload is maximized) for a particular mission  $\Delta V$ . He demonstrated this by differentiating the above expression with respect to the parameter  $c/V_c$  and setting it equal to zero. The result is the optimization parameter:

$$\exp \frac{\Delta V}{c} = \frac{1}{1-\lambda} = \frac{\Delta V}{2c} \left[ 1 + \left( \frac{V_c}{c} \right)^2 \right] + 1 \quad (11)$$

Shepherd then took the classical distance equation for a single burn, continuously accelerating rocket (neglecting gravity losses):

$$S(t) = V_0 t + \frac{cT}{\lambda} \left[ \left( 1 - \lambda \frac{t}{T} \right) \ln \left( 1 - \lambda \frac{t}{T} \right) - \left( 1 - \lambda \frac{t}{T} \right) + 1 \right] \quad (12)$$

and solved for total distance traveled in time T (assuming zero initial velocity):

$$S = \frac{cT}{\lambda} [(1-\lambda) \ln(1-\lambda) + \lambda] = cT \left[ \frac{\ln \left( \exp \frac{-\Delta V}{c} \right)}{\exp \frac{\Delta V}{c} - 1} + 1 \right] \quad (13)$$

By substituting the optimization parameter Eq. (11) for the exponential term in the denominator of Eq. (13), we have:

$$\frac{S}{cT} = \frac{1 - \left(\frac{c}{V_c}\right)^2}{1 + \left(\frac{c}{V_c}\right)^2} \quad (14)$$

At this point, Shepherd concluded with discussion of the implications of this acceleration-only distance equation.

What is of primary interest is to rewrite Shepherd's single burn flyby mission distance equation into a form where trip time is explicitly a function of distance,  $I_{sp}$ , and  $\alpha$ . Expressing  $c$  as  $g_c I_{sp}$  and recalling the definition of  $V_c$ , Eq.(14) can be rewritten as a quadratic in  $T$ :

$$T^2 - \left[ \frac{S}{g_c I_{sp}} + \frac{g_c I_{sp}^2}{2\alpha\eta} \right] T - \frac{S I_{sp}}{2\alpha\eta} = 0 \quad (15)$$

Solving for trip time via the quadratic formula:

$$T_{1,2} = \frac{1}{2} \left[ \frac{S}{g_c I_{sp}} + \frac{g_c I_{sp}^2}{2\alpha\eta} \right] \pm \frac{1}{2} \sqrt{\left( \frac{g_c I_{sp}^2}{2\alpha\eta} \right)^2 + \frac{3S I_{sp}}{\alpha\eta} + \left( \frac{S}{g_c I_{sp}} \right)^2} \quad (16)$$

Where only the positive radical term is meaningful.

Shepherd demonstrated that  $c/V_c$  has limits of  $\approx 0.505 \leq c/V_c \leq 1.0$ , where the lower limit corresponds to a zero payload ratio and the upper limit corresponds to a payload ratio of unity. These limits are a manifestation of imposing the optimization parameter on the rocket equation. Since the payload ratio is a function of  $c/V_c$  and  $\Delta V/c$ , and  $\Delta V/c$  is a function of only  $c/V_c$ , specifying  $c/V_c$  is equivalent to specifying a payload ratio. Since  $c/V_c$  is solely a function of  $T$ ,  $I_{sp}$ , and  $\alpha$ , payload ratio can replace one of these three variables. For example, it was convenient to express the limiting  $\alpha = f(I_{sp}, S)$  for the zero payload ratio case:

$$\frac{c}{V_c} = \frac{g_c I_{sp}}{\sqrt{2g_c \eta \alpha T}} \approx 0.505 \quad \text{for} \quad \frac{M_{pay}}{M_i} = 0 \quad (17)$$

$$\text{therefore} \quad \alpha_{lim} = \frac{g_c I_{sp}^2}{2(0.505)^2 \eta T} \quad (18)$$

Substituting Eq. (18) into Eq. (16) and selecting the meaningful root results in an equation for the limiting trip time:

$$T_{\lim} = 1.68 \frac{S}{g_c I_{sp}} \quad (19)$$

Thus the minimum trip time for a single burn flyby mission with negligible payload and optimal  $\alpha$  varies linearly with distance and inversely with  $I_{sp}$ . The optimal  $\alpha$  can then be calculated from Eq. (18).

### Derivation of the Solution to the Two-Burn Rendezvous Problem

Unlike the one-burn flyby mission, the rendezvous mission would include a deceleration burn to reduce the sun-relative radial velocity to zero by the time of arrival at the destination planet. The rendezvous mission also presupposed the utilization of in situ resources at the destination planet to refuel the propulsion system, since the vehicle would carry only enough propellants for a one way trip. A modified rocket equation to model the deceleration burn is first needed. Shepherd's time dependent form of the constant mass flow/constant thrust rocket equation for the acceleration burn ( $0 < t \leq t_1$ ) was:

$$V_1(t) = V_0 - g_c I_{sp} \ln \left[ 1 - \lambda_1 \frac{t}{t_1} \right] \quad (20)$$

Therefore, a new deceleration burn ( $t_1 \leq t < t_2$ ) velocity function can be modeled similarly as:

$$V_2(t) = V_1 + g_c I_{sp} \ln \left[ 1 - \lambda_2 \frac{t-t_1}{t_2-t_1} \right] \quad (21)$$

$V_1$  is the velocity at the turn around point. This point is where the vehicle terminates acceleration, rotates 180 degrees, and commences the deceleration burn. The logarithmic term, therefore, is a negative or "braking"  $\Delta V$ . The distance and acceleration expressions for the deceleration phase are easily derivable. To integrate the logarithmic term, the following substitution is convenient:

$$\text{let } u \equiv 1 - \lambda_2 \left( \frac{t-t_1}{t_2-t_1} \right) \quad \text{then } \frac{du}{dt} = \frac{-\lambda_2}{t_2-t_1} \quad (22)$$

$$S_2(t) = V_1 \int_{t_1}^t dt + g_c I_{sp} \int_{t_1}^t \ln \left[ 1 - \lambda_2 \left( \frac{t-t_1}{t_2-t_1} \right) \right] dt = V_1(t-t_1) + g_c I_{sp} \frac{t_2-t_1}{-\lambda_2} \int_1^u \ln u \, du \quad (23)$$

Completing the integration results in an equation for deceleration distance as a function of time:

$$S_2(t) = V_1(t-t_1) - \frac{g_c J_{sp}(t_2-t_1)}{\lambda_2} \left[ \left( 1 - \lambda_2 \frac{t-t_1}{t_2-t_1} \right) \ln \left( 1 - \lambda_2 \frac{t-t_1}{t_2-t_1} \right) - \left( 1 - \lambda_2 \frac{t-t_1}{t_2-t_1} \right) + 1 \right] \quad (24)$$

Taking the time derivatives of the velocity equations result in the acceleration and deceleration equations as a function of time:

$$A_1(t) = \frac{dV_1(t)}{dt} = \frac{g_c J_{sp} \lambda_1}{t_1 \left[ 1 - \lambda_1 \frac{t}{t_1} \right]} \quad (25)$$

$$A_2(t) = \frac{dV_2(t)}{dt} = \frac{-g_c J_{sp} \lambda_2}{(t_2-t_1) \left[ 1 - \lambda_2 \frac{t-t_1}{t_2-t_1} \right]} \quad (26)$$

A representative plot of the position, velocity, and acceleration expressions for an example mission are illustrated in figure 2.

A total distance equation can be formed and the variables  $S_1$ ,  $S_2$ ,  $V_1$ ,  $t_1$ ,  $t_2$ ,  $\lambda_1$ , and  $\lambda_2$  replaced with more convenient variables by inserting known relations and constraints. The total distance traveled is merely:

$$S = S_1 + S_2 \quad (27)$$

The magnitude of the velocity at end of the acceleration phase is:

$$V_1 = -g_c J_{sp} \ln(1 - \lambda_1) \quad (28)$$

It is reasonable to assume that the magnitude of the change of velocities will be equal:

$$-g_c J_{sp} \ln \left[ 1 - \lambda_1 \left( \frac{t_1}{t_1} \right) \right] = -g_c J_{sp} \ln \left[ 1 - \lambda_2 \left( \frac{t_2-t_1}{t_2-t_1} \right) \right] \quad (29)$$

Continuous Acceleration/Deceleration Mission  
Constant Thrust, Isp, & Power  
Example: Isp=50,000 sec and Alpha= 100 kW/kg

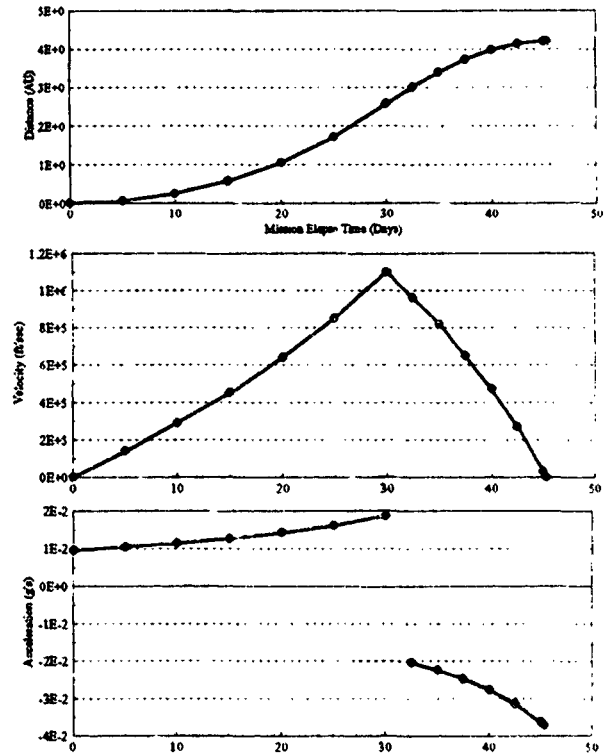


Figure 2: Position, Velocity, & Acceleration



Thus, the stage propellant ratios are equal.

$$\lambda_1 = \frac{M_{p1}}{M_i} = \lambda_2 = \frac{M_{p2}}{M_i - M_{p1}} = \lambda \quad (30)$$

Adding the two distance expressions (Eqs. (12) and (24), evaluated at their respective total burn times) and substituting Eqs. (28) and (30) results in an equation for total distance traveled:

$$S = g_c I_{sp} t_1 \left[ \left( \frac{1-\lambda}{\lambda} \right) \ln(1-\lambda) + 1 \right] - g_c I_{sp} (t_2 - t_1) \ln(1-\lambda) - g_c I_{sp} (t_2 - t_1) \left[ \left( \frac{1-\lambda}{\lambda} \right) \ln(1-\lambda) + 1 \right] \quad (31)$$

The total distance traveled, S, is the straight line distance between the orbits of the two planets. It may be set to the difference between the radii (shortest trip time) or any other distance up to the opposition of the planets.

Note that the deceleration phase propellant ratio with respect to the initial mass can be written:

$$\lambda_2 = \frac{M_{p2}}{M_i - M_{p1}} = \frac{\frac{M_{p2}}{M_i}}{1 - \frac{M_{p1}}{M_i}} = \frac{\frac{M_{p2}}{M_i}}{1 - \lambda_1} \quad \text{thus} \quad \frac{M_{p2}}{M_i} = \lambda(1-\lambda) \quad (32)$$

A relationship can be found relating the two burn times by noting that the mass flow rates are equal and that the total mission time is merely the sum of the two burn phases:

$$\frac{M_{p1}}{t_1} = \frac{M_{p2}}{t_2 - t_1} \quad \text{thus} \quad \frac{t_1}{t_2 - t_1} = \frac{\frac{M_{p1}}{M_i}}{\frac{M_{p2}}{M_i}} = \frac{1}{1-\lambda} \quad (33)$$

After replacing  $t_1$  in Eq. (31) with Eq. (33), and noting that  $t_2$  is the total trip time T, the logarithmic terms cancel and the distance equation reduces to:

$$S = \frac{g_c I_{sp} T \lambda}{2 - \lambda} \quad (34)$$

This is quite different from the acceleration-only distance Eq. (13), which is a logarithmic function of  $\lambda$ . The implication is that attempting to substitute the optimization parameter Eq. (11) (with its exponential function of  $\Delta V/c$ ) into the distance equation, as was done in the one-burn flyby case, will not result in a simple function of distance,  $I_{sp}$ , and  $\alpha$  (since there will not be a clean cancellation of the intermediate variable  $\Delta V/c$ ). The distance and optimization equations for the two-burn rendezvous case (as well as the four-burn round trip case as we shall see) are therefore coupled by the dependent variable  $\lambda$  and must be solved by iteration.

It was found to be easiest to have  $\lambda$  serve as the iteration parameter. Remember that the  $\lambda$  used above was defined as a stage propellant ratio. It must first be replaced with the equivalent expression for the total propellant ratio, which is implicit in the optimizing parameter equation

$$1 - \exp^{-\frac{\Delta V}{g_c I_{sp}}} = \lambda_{total} = \frac{M_{p1} + M_{p2}}{M_i} = \frac{M_{p1}}{M_i} + \frac{M_{p2}}{M_i} = \lambda_{stage} + \lambda_{stage}(1 - \lambda_{stage}) \quad (35)$$

This quadratic in  $\lambda_{stage}$  has a solution of:

$$\lambda_{stage} = 1 \pm \sqrt{1 - \lambda_{total}} \quad (36)$$

For the cases of interest here, where  $0.505 \leq c/V_c \leq 1.0$ , only the negative root term is meaningful. Using this expression to replace  $\lambda_{stage}$  with  $\lambda_{total}$  in Eq. (34) yields:

$$S = \frac{g_c I_{sp} T(1 - \sqrt{1 - \lambda_{total}})}{(1 + \sqrt{1 - \lambda_{total}})} \quad \text{or} \quad \lambda_{total} = 1 - \left(1 - \frac{2S}{g_c I_{sp} T + S}\right)^2 \quad (37)$$

The optimizing parameter Eq. (11) can be expressed as a function of  $\lambda_{total}$ ,  $I_{sp}$ , and  $\alpha$  by recalling the definition of  $\lambda$  from the rocket equation and the definition of  $c/V_c$  as a function of  $I_{sp}$ ,  $\alpha$ , and  $T$  (Eq. (8)). Substituting these relations, the optimizing parameter can be rewritten as:

$$T = \frac{g_c I_{sp}^2}{2\eta\alpha} \left[ \frac{2\lambda_{total}}{(1 - \lambda_{total}) \ln(1 - \lambda_{total})^{-1}} - 1 \right] \quad (38)$$

Eqs. (37) and (38) are sufficient to solve the two-burn rendezvous problem. By choosing an initial value of  $\lambda_{total}$  and substituting into Eq. (38), an initial trip time can be found. (From Shepherd, an initial (maximum)  $\lambda_{total}$  value of 0.796812 is suggested. <sup>4</sup>) The trip time can then be substituted into Eq. (37) to find a new value of  $\lambda_{total}$ . Repeated iteration using the average of the  $\lambda_{total}$  values as the new  $\lambda_{total}$  was found to converge quite rapidly in most cases.

Once a converged solution is found, several dependent variables of interest can easily be calculated. The first burn propellant ratio and  $c/V_c$  can be calculated from Eqs. (36) and (8) respectively. The structure, payload, and initial & final thrust-to-weight ratios are then given by:

$$\frac{M_s}{M_i} = \left( \frac{M_p}{M_i} \right) \lambda_{total} = \left( \frac{c}{V_c} \right)^2 \lambda_{total} \quad (39)$$

$$\frac{M_{pay}}{M_i} = 1 - \left( \frac{M_p}{M_i} \right) - \left( \frac{M_s}{M_i} \right) = 1 - \lambda_{total} - \left( \frac{c}{V_c} \right)^2 \lambda_{total} \quad (40)$$

$$\frac{F}{W_i} = \frac{1}{g} \frac{c \lambda_{stage}}{t_1} = \frac{g_c I_{sp}}{g T} \lambda_{stage} (2 - \lambda_{stage}) \quad (41)$$

$$\frac{F}{W_f} = \frac{1}{g} \frac{-c \lambda_{stage2}}{(t_2 - t_1)(1 - \lambda_{stage2})} = \frac{g_c I_{sp}}{g T} \frac{\lambda_{stage}(2 - \lambda_{stage})}{(1 - \lambda_{stage})^2} \quad (42)$$

As in the single burn acceleration case, payload ratios can be specified and the limiting  $I_{sp}$ 's and  $\alpha$ 's can be calculated. The value of the parameter  $c/V_c$  for each corresponding value of payload ratio must be found first. This can be accomplished by solving the rocket equation (rewritten in terms of mass ratios) and the optimization parameter simultaneously after specifying the desired payload fraction. These equations are coupled, but were found to be easily solved by iteration. Shepherd had provided an approximate value of 0.505 for the zero-payload case. Alternate and more exact values were desired for analysis purposes. For 0% and 25% payload fractions, the values of  $c/V_c$  were calculated to be 0.504976295 and 0.736886943 respectively. The value of  $\lambda_{stage}$  can then be found by inserting the optimization parameter Eq. (11) into the rewritten rocket equation Eq. (10) (replacing the exponential term), substituting Eq. (36) for the equivalent  $\Delta V/c$  term, and solving for  $\lambda_{stage}$ . The result is:

$$\lambda_{stage_{opt}} = 1 - \exp \left[ \frac{\left[ \left( \frac{M_{pay}}{M_i} \right) - 1 \right] \left( \frac{c}{V_c} \right)^2}{\left[ \left( \frac{c}{V_c} \right)^2 + 1 \right] \left[ \left( \frac{c}{V_c} \right)^2 + \left( \frac{M_{pay}}{M_i} \right) \right]} \right] \quad (43)$$

Once  $\lambda_{\text{max}}$  is known, the optimum travel time can be easily calculated from Eq. (34). The optimum  $\alpha$  can then be calculated from Eq. (8). At this point, the two-burn problem and the parameters of interest are solved. These relations can be manipulated by hand or easily assembled into a short computer routine.

#### Derivation of the Solution to the Four-Burn Round Trip Problem

A similar set of equations were derived for a four-burn round trip mission, where the spacecraft would carry all of its propellant without the use of *in situ* propellant refueling. The equations were found to be similar to the two-burn rendezvous mission. The two-burn rendezvous mission can be thought of as the return leg of a round trip mission. The velocity equations are of the same form as the two-burn mission, as are the acceleration equations. Since the outgoing and returning distances will be assumed to be equal

$$S_{\text{outgoing}} = S_{\text{returning}} = \frac{g_c I_{sp} T_{\text{outgoing}} \lambda_1}{2 - \lambda_1} = \frac{g_c I_{sp} T_{\text{returning}} \lambda_3}{2 - \lambda_3} \quad (44)$$

$$\text{thus : } \frac{T_{\text{outgoing}}}{T_{\text{return}}} = \frac{\lambda_3(2 - \lambda_1)}{\lambda_1(2 - \lambda_3)} \quad (45)$$

The relationships between trip times and  $\lambda$ 's must be found. Since the mass flow rates are constant:

$$\frac{T_{\text{outgoing}}}{T_{\text{return}}} = \frac{\frac{M_{p_1} + M_{p_2}}{M_i}}{\frac{M_{p_3} + M_{p_4}}{M_i}} = \frac{\lambda_1 + \lambda_1(1 - \lambda_1)}{\lambda_3(1 - \lambda_1)^2 + \lambda_3(1 - \lambda_3)(1 - \lambda_1)^2} \quad (46)$$

Equating Eqs. (45) and (46),  $\lambda_1$  and  $\lambda_3$  are thus related by:

$$\lambda_3 = \frac{\lambda_1}{1 - \lambda_1} \quad (47)$$

An expression for  $\lambda$ 's in terms of  $\lambda_{\text{total}}$  can be found since:

$$\lambda_{\text{total}} = \frac{M_{p_1}}{M_i} + \frac{M_{p_2}}{M_i} + \frac{M_{p_3}}{M_i} + \frac{M_{p_4}}{M_i} = \lambda_1 + \lambda_1(1 - \lambda_1) + \lambda_3(1 - \lambda_1)^2 + \lambda_3(1 - \lambda_3)(1 - \lambda_1)^2 \quad (48)$$

Substituting Eq. (47) into Eq. (48) results in a quadratic expression for  $\lambda_1$  in terms of  $\lambda_{total}$ :

$$\lambda_1^2 - \lambda_1 + \frac{\lambda_{total}}{4} = 0 \quad \text{with roots of: } \lambda_1 = \frac{1 \pm \sqrt{1 - \lambda_{total}}}{2} \quad (49)$$

Only the negative radical term is meaningful. Notice that this first burn propellant fraction is half the expression found in the two-burn rendezvous case (Eq. (36)).

Finally, the outgoing trip time can be expressed as a fraction of the total round trip time by again noting that mass flow rate is constant:

$$\frac{T_{outgoing}}{T} = \frac{\frac{M_{p_1} + M_{p_2}}{M_i}}{\frac{M_{p_1} + M_{p_2} + M_{p_3} + M_{p_4}}{M_i}} = \frac{\lambda_1(2 - \lambda_1)}{\lambda_1(2 - \lambda_1) + \lambda_1(2 - 3\lambda_1)} \quad (50)$$

Substituting Eq. (50) into the round trip distance Eq. (44):

$$S_{roundtrip} = 2S_{rendezvous} = 2g_c I_{sp} T_{outgoing} \left( \frac{\lambda_1}{2 - \lambda_1} \right) = \frac{g_c I_{sp} T}{2} \left( \frac{\lambda_1}{1 - \lambda_1} \right) \quad (51)$$

As was done for the rendezvous case, this can be solved for  $\lambda_{total}$  using Eq. (49). For convenience, the distance can be written in terms of  $S_{rendezvous}$  rather than  $S_{roundtrip}$ :

$$2S_{rendezvous} = \frac{g_c I_{sp} T (1 - \sqrt{1 - \lambda_{total}})}{2(1 + \sqrt{1 - \lambda_{total}})} \quad \text{or} \quad \lambda_{total} = 1 - \left( 1 - \frac{8S_{rendezvous}}{g_c I_{sp} T + 4S_{rendezvous}} \right)^2 \quad (52)$$

Using this equation and the optimization parameter Eq. (38), the round trip mission trip time can be calculated in the same way as was done for the rendezvous case. The  $c/V_c$ , structure, and payload ratios are calculated the same as in the rendezvous mission (Eqs. (8), (39), and (40)). The optimum fixed payload (Eq. (43)) is reduced by a factor of one half as per Eq. (49). The thrust-to-weight ratios are:

$$\frac{F}{W_i} = \frac{1}{g} \frac{c \lambda_{stage}}{t_1} = 4 \frac{g_c I_{sp}}{g T} \lambda_{stage} (1 - \lambda_{stage}) \quad (53)$$

$$\frac{F}{W_f} = \frac{1}{g} \frac{-c \lambda_{stage^4}}{(t_4 - t_3)(1 - \lambda_{stage^4})} = -4 \frac{g_c I_{sp}}{g T} \frac{\lambda_{stage} (1 - \lambda_{stage})}{(1 - 2\lambda_{stage})^2} \quad (54)$$

## RESULTS

Iterations on the trip time/optimization parameter equations were well behaved and converged rapidly. Although sample hand calculations were not found to be overly burdensome, a simple computer routine was written to calculate almost all of the following data. As an example, rendezvous and round trip missions to Jupiter were calculated. After this data was generated, the same mission-space parameters were calculated using a large, numerical integration trajectory optimization routine to independently verify the data and establish the limits of validity of this approach. The position of Jupiter was assumed to be at minimum distance from Earth (4.203 AU) at departure. Round trip missions assumed no dwell time at the destination and no significant relative motion of the planets. Ranges of  $I_{sp}$  from 50,000 to 200,000 seconds and  $\alpha$  from 100 to 1000 kW/kg were representative of what a propulsion system must operate at to perform multi-week/multi-month transfers to this nearest of major planets. Overall propulsion system efficiency was set to unity. Figure 3 illustrates rendezvous mission trip time as a function of both  $I_{sp}$  and  $\alpha$ . A rendezvous trip time to Jupiter of six weeks, for example, will require propulsion technologies with  $I_{sp}$ 's of at least 50,000 seconds and  $\alpha$ 's of at least 100 kW/kg. For constant  $I_{sp}$ , increasing  $\alpha$  always decreases trip time since the engine jet power-out increases (and/or the structure mass decreases). For constant  $\alpha$ , increasing  $I_{sp}$  always lengthens trip time as the propulsion system begins to act more like a low thrust electric device.  $I_{sp}$ 's beyond the 50,000 to 100,000 seconds levels are excessive for interplanetary  $\Delta V$ 's unless a corresponding (significant) improvement in  $\alpha$  also occurs. A propulsion system engineer, planning to conduct proof of concept experiments, can use these observations to readily trade-off work on one parameter for the other depending on the judged engineering difficulty. Propulsion technology research can thus be guided by the knowledge of the relative merit of pursuing greater thrust vs. greater power.

Using the fixed payload mass fraction equations, payload contours can be superimposed onto the  $I_{sp}$  &  $\alpha$  plots as a means of measuring performance. Two performance levels were calculated, a 0% (fastest trip time) and a 25% payload mass fraction. Other desired payload mass fractions between these two values can be approximated by interpolation. Greater values should not be estimated in this way due to the non-linearity of the function as  $c/V_e$  increases. For trip times of approximately six weeks, the maximum payload mass fractions for  $I_{sp}$ 's between 50,000 and 100,000 seconds were 3% to 25% respectively. To halve trip times (three weeks),  $I_{sp}$ 's would have to be doubled and  $\alpha$ 's would have to be improved by an order of magnitude.

Figure 4 illustrates the results for round trip missions, using similar ranges of  $I_{sp}$  and  $\alpha$ . In the round trip mission, the spacecraft departing from Earth carries all the fuel required to perform the entire mission. By observation, the data illustrates that (like terrestrial travel) refueling at a destination is a compelling way to operate. For example, a zero payload/ 50,000 second  $I_{sp}$  vehicle that refuels at a destination would take half the time (2 X 40 days) to perform the same mission as a same initial weight vehicle carrying all of its propellant from the start. This enhances the attractiveness of travel to the major outer planets due to their abundant supply of potential fuels and propellants such as hydrogen, deuterium, and helium-3. These materials are plentiful in both free molecular form in planetary atmospheres and in bound forms on the surfaces of their moons. Major planets would not only be destinations, but transportation nodes as well, supplying resources for fuels/propellants and human consumption.

Figure 5 and Table 1 contain additional rendezvous mission data: mass fractions, trip times, and thrust-to-weight data for  $I_{sp} = 50,000$  seconds plotted as a function of  $\alpha$ . For fixed  $I_{sp}$ , as  $\alpha$  increases, the amount of propellant needed increases at the expense of payload fraction. The structure ratio reaches a maximum at approximately 26% as Shepherd's work had predicted<sup>4</sup>. As trip time is reduced by increasing  $\alpha$ , payload ratio becomes vanishingly small. Thrust-to-weight ratios depart from near constant values as  $\alpha$  increases. Figure 6 and Table 2 contain similar data for round trip missions, plotted as a function of  $I_{sp}$ .

## Example: Mission to Jupiter (Rendezvous)

Trip Time vs. Isp & Alpha

(Maximized payload ratio, minimized travel time)

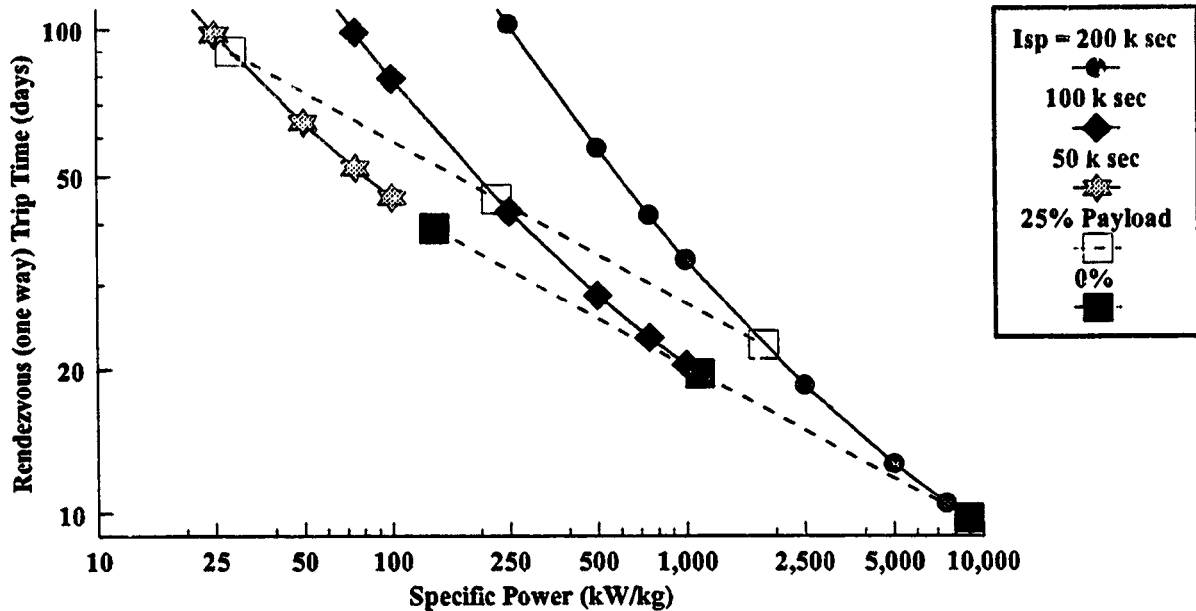


Figure 3. Rendezvous Mission TripTime

## Example: Mission to Jupiter (Round Trip)

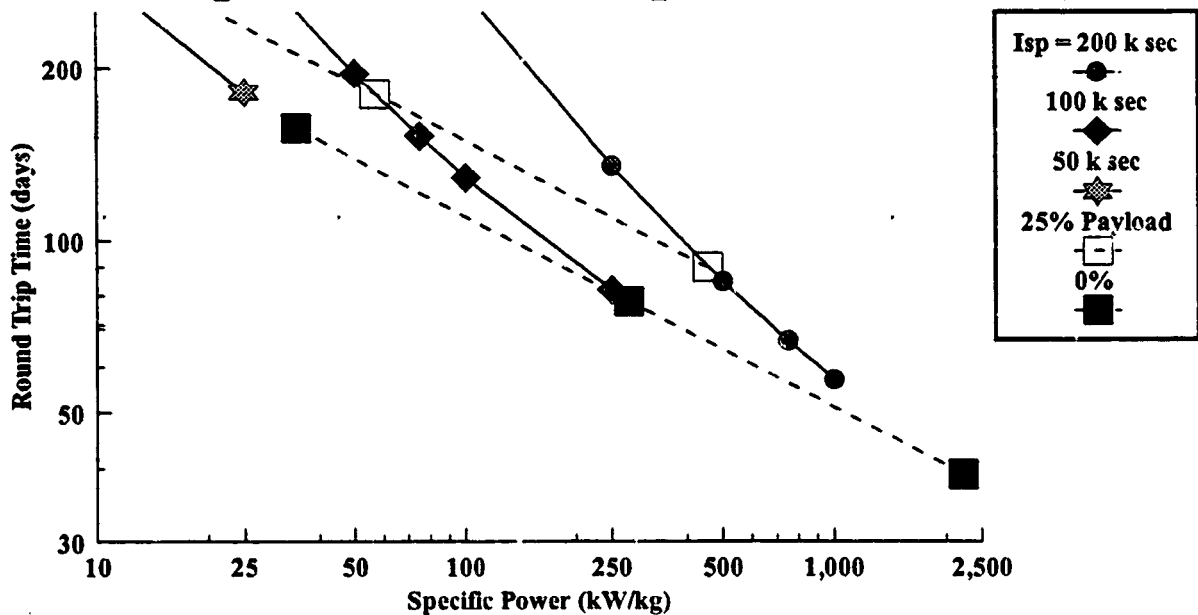


Figure 4. Round Trip Mission TripTime

Figure 5. Selected Mission Data  
**Example: Mission to Jupiter (Rendezvous)**  
 Mass Ratio , Trip Time, & Thrust -to-Weight vs. Specific Power

Isp = 50,000 Sec

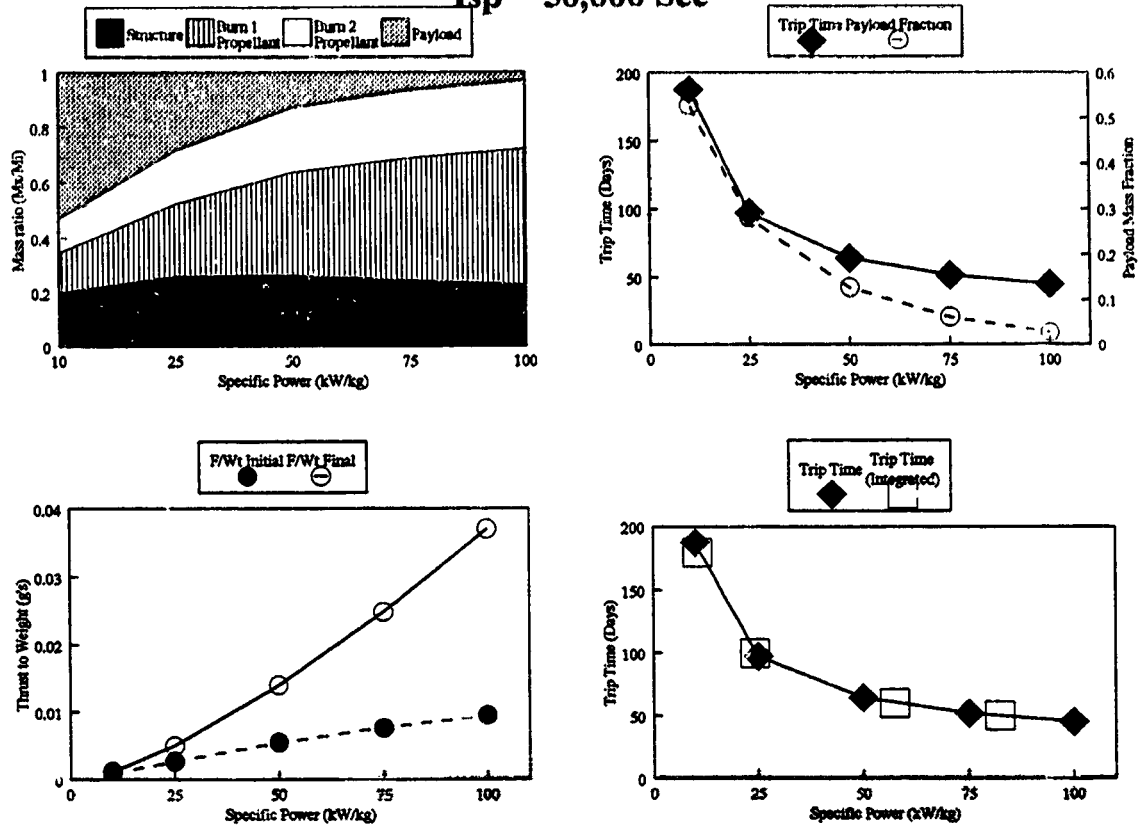


Table 1. Selected Mission Data

Isp = 50,000 Sec

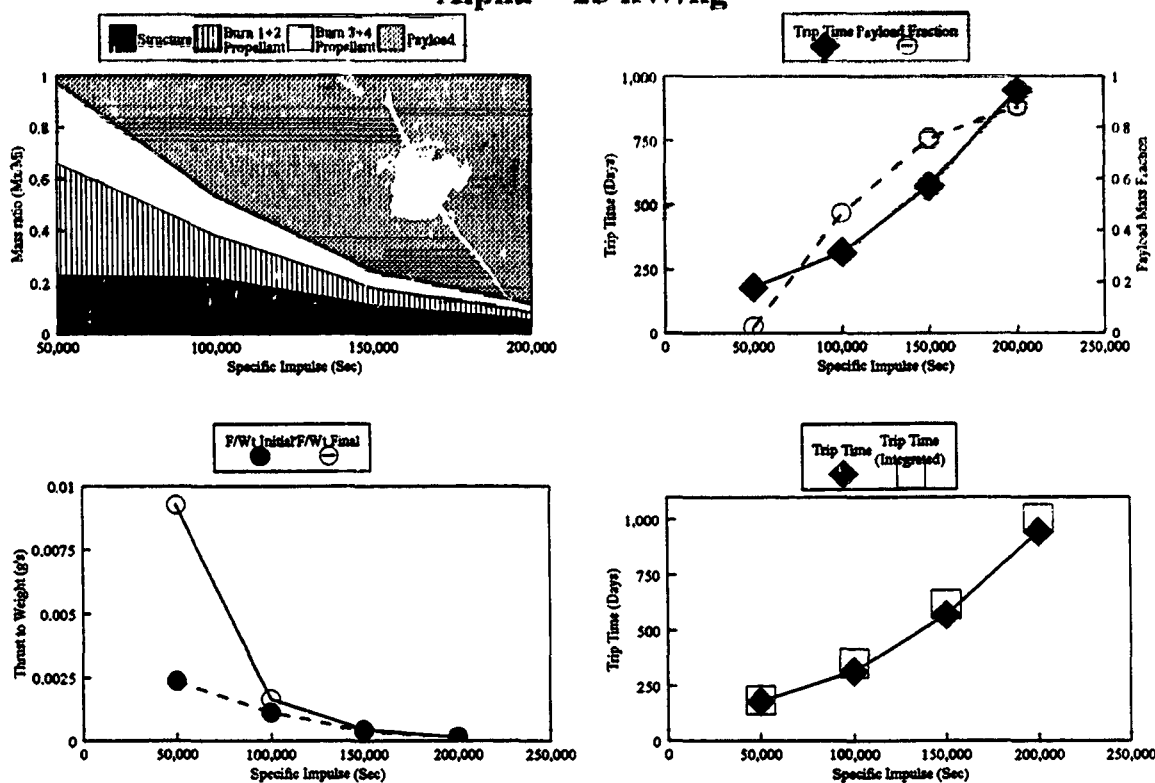
| Specific Power (kW/kg) | Time (days) | Payload Ratio | Structure Ratio | Total Prop Ratio | 1st Burn Prop Ratio | c/Vc  | F/Wt Initial Accel(g/s) | F/Wt Final Decel(g/s) |
|------------------------|-------------|---------------|-----------------|------------------|---------------------|-------|-------------------------|-----------------------|
| 10                     | 187.9       | 0.528         | 0.200           | 0.271            | 0.146               | 0.860 | 0.835E-3                | 0.115E-2              |
| 25                     | 97.7        | 0.282         | 0.260           | 0.458            | 0.264               | 0.754 | 0.271E-2                | 0.500E-2              |
| 50                     | 64.4        | 0.128         | 0.263           | 0.609            | 0.375               | 0.657 | 0.548E-2                | 0.140E-1              |
| 75                     | 52.0        | 0.063         | 0.246           | 0.691            | 0.444               | 0.597 | 0.769E-2                | 0.249E-1              |
| 100                    | 45.3        | 0.028         | 0.228           | 0.744            | 0.494               | 0.554 | 0.950E-2                | 0.371E-1              |

Alpha = 100 kW/kg

| Specific Impulse (Sec) | Time (days) | Payload Ratio | Structure Ratio | Total Prop Ratio | 1st Burn Prop Ratio | c/Vc  | F/Wt Initial Accel(g/s) | F/Wt Final Decel(g/s) |
|------------------------|-------------|---------------|-----------------|------------------|---------------------|-------|-------------------------|-----------------------|
| 50,000                 | 45.3        | 0.028         | 0.228           | 0.744            | 0.494               | 0.554 | 0.950E-2                | 0.371E-1              |
| 100,000                | 79.3        | 0.468         | 0.219           | 0.313            | 0.171               | 0.837 | 0.457E-2                | 0.665E-2              |
| 150,000                | 143.1       | 0.758         | 0.113           | 0.129            | 0.067               | 0.934 | 0.157E-2                | 0.180E-2              |
| 200,000                | 236.5       | 0.882         | 0.057           | 0.061            | 0.031               | 0.969 | 0.595E-3                | 0.634E-3              |



**Figure 6. Selected Mission Data**  
**Example: Mission to Jupiter (Round Trip)**  
**Mass Ratio , Trip Time, & Thrust -to-Weight vs. Specific Impulse**  
**Alpha = 25 kW/kg**



**Table 2. Selected Mission Data**

**Isp = 200,000 Sec**

| Specific Power (kW/kg) | Time (days) | Payload Ratio | Structure Ratio | Total Prop Ratio | 1st Burn Prop Ratio | c/Vc  | F/Wt Initial Accel(g's) | F/Wt Final Decel(g's) |
|------------------------|-------------|---------------|-----------------|------------------|---------------------|-------|-------------------------|-----------------------|
| 25                     | 946.1       | 0.882         | 0.057           | 0.061            | 0.015               | 0.969 | 0.149E-3                | 0.158E-3              |
| 50                     | 499.5       | 0.788         | 0.100           | 0.112            | 0.029               | 0.943 | 0.520E-3                | 0.585E-3              |
| 75                     | 349.7       | 0.711         | 0.132           | 0.156            | 0.041               | 0.920 | 0.103E-2                | 0.123E-2              |
| 100                    | 274.2       | 0.647         | 0.158           | 0.195            | 0.051               | 0.900 | 0.165E-2                | 0.204E-2              |
| 250                    | 134.8       | 0.407         | 0.236           | 0.357            | 0.099               | 0.812 | 0.614E-2                | 0.956E-2              |
| 500                    | 84.8        | 0.228         | 0.266           | 0.507            | 0.149               | 0.724 | 0.138E-1                | 0.280E-1              |
| 750                    | 66.7        | 0.140         | 0.264           | 0.595            | 0.182               | 0.666 | 0.207E-1                | 0.510E-1              |
| 1000                   | 57.1        | 0.090         | 0.255           | 0.655            | 0.206               | 0.624 | 0.266E-1                | 0.771E-1              |

**Alpha = 25 kW/kg**

| Specific Impulse (Sec) | Time (days) | Payload Ratio | Structure Ratio | Total Prop Ratio | 1st Burn Prop Ratio | c/Vc  | F/Wt Initial Accel(g's) | F/Wt Final Decel(g's) |
|------------------------|-------------|---------------|-----------------|------------------|---------------------|-------|-------------------------|-----------------------|
| 50,000                 | 181.1       | 0.028         | 0.228           | 0.744            | 0.247               | 0.554 | 0.238E-2                | 0.927E-2              |
| 100,000                | 317.2       | 0.468         | 0.219           | 0.313            | 0.086               | 0.837 | 0.114E-2                | 0.166E-2              |
| 150,000                | 572.6       | 0.758         | 0.113           | 0.129            | 0.033               | 0.934 | 0.392E-3                | 0.450E-3              |
| 200,000                | 946.1       | 0.882         | 0.057           | 0.061            | 0.015               | 0.969 | 0.149E-3                | 0.158E-3              |

**Table 3. Equation Summary**

|                        | 1 Burn<br>Flyby | 2 Burn<br>Rendezvous | 4 Burn<br>Round Trip |
|------------------------|-----------------|----------------------|----------------------|
| Total<br>Distance      | (13)            | (37)                 | (52)                 |
| Optimized<br>Trip Time | (16)            | (37) & (38)          | (52) & (38)          |
|                        | Position        | Velocity             | Acceleration         |
| Acceleration<br>Phase  | (12)            | (20)                 | (25)                 |
| Deceleration<br>Phase  | (24)            | (21)                 | (26)                 |

To check the validity of these results, a series of integrated computer runs were performed using an existing, high fidelity interplanetary trajectory optimization routine<sup>5</sup>. A few of the results of these runs are superimposed onto the data generated and are shown in the last plot, Figures 5 and 6. It was found that at these high power levels and with accelerations well in excess of the local acceleration due to solar gravity, trip time results agreed within 1% for rendezvous and 6% for round trip missions. The larger error in round trip times was noticeable in cases where travel times were a considerable fraction (or greater) of one Earth year. In these cases, the integration program significantly altered the outgoing trip to accommodate the change in relative position of the Earth upon return. Since the approach outlined in this paper does not account for change in the relative positions of the planets implicitly, closer agreement for round trip missions is limited to the time scales similar to those of rendezvous missions. Table 3 provides an overall summary of pertinent trip time relations and equations of motion for the analyst.

## CONCLUSIONS

A simple, analytic approximation was shown to provide a means to readily calculate trip time and performance of propulsion systems of very high  $I_{sp}$  &  $\alpha$  for human interplanetary space missions. Simultaneous solution of two equations was shown to be all that was necessary without having to resort to complex computer programs to integrate trajectories. The simplifying assumptions of one dimensional motion, field-free space, constant thrust/ $I_{sp}/\alpha$ , and near straight line (radial) trajectories between the planets permitted data of sufficient accuracy to be generated, providing insight into the relationships between distance,  $I_{sp}$ , and  $\alpha$ . An optimizing parameter was included to maximize performance and minimize trip time. Example data was provided for rendezvous and round trip missions to Jupiter, illustrating trip times and payload mass fractions for a wide range of  $I_{sp}$  and  $\alpha$ . Results indicated that a propulsion system technology must be capable of  $I_{sp}$ 's of at least 50,000 seconds and  $\alpha$ 's of at least 100 kW/kg to travel to Jupiter in 6 weeks (rendezvous) and still allow for a 3% payload mass fraction. Overall results were shown to agree within 1% and 6% (rendezvous and round trip, respectively) of data independently generated, verifying that for preliminary analysis, a technology planner need not resort to large, high fidelity computer programs that require expert operators in order to accurately characterize the propulsion system parameter-space. Comparison of data between two-burn rendezvous and four-burn round trip missions illustrated the extreme attractiveness of in situ refueling at the destination planet. Few propulsion technologies have been identified that are expected to be capable of providing such demanding  $I_{sp}$ 's and  $\alpha$ 's. Inertial confinement fusion systems may be capable of  $I_{sp}$ 's between 50,000 to 270,000 seconds and  $\alpha$ 's as large as 100 kW/kg<sup>6</sup>. Antiproton-catalyzed fusion and antiproton annihilation are other potential space propulsion concepts that are expected to operate at or beyond this regime.

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Discussions with Dr. Stanley Borowski of the NASA Lewis Research Center and Professor Terry Kammash of the University of Michigan were invaluable in the formulation of this approach. Their earlier work deriving selected equations using an alternate approach to this problem provided the necessary guidance and confirmation of the validity of this method<sup>7,8</sup>. Special thanks to Leon Gefert and Leonard Dudzinski of the NASA Lewis Research Center for the numerous integrated trajectories that were run to validate the equations.

## NOTATION

|            |                                                            |
|------------|------------------------------------------------------------|
| A          | acceleration (ft/sec**2)                                   |
| $\alpha$   | specific power (ft lbf/(sec lbm))                          |
| c          | exhaust velocity (ft/sec) = $g_c I_{sp}$                   |
| F          | thrust (lbf)                                               |
| g          | gravitational acceleration: 32.1739 (ft/sec**2)            |
| $g_c$      | conversion constant: 32.1739 (lbm/lbf)(ft/sec**2)          |
| $I_{sp}$   | specific impulse (lbf sec/lbm) = thrust per mass flow rate |
| $\lambda$  | propellant mass fraction ( $M_p/M_t$ )                     |
| M          | mass (lbm)                                                 |
| $\eta$     | overall propulsion system efficiency                       |
| P          | jet power (ft lbf/sec)                                     |
| S          | distance (ft)                                              |
| t          | time (sec)                                                 |
| T          | trip time (sec)                                            |
| $V_o$      | characteristic velocity (ft/sec)                           |
| V          | velocity (ft/sec)                                          |
| $\Delta V$ | velocity increment (ft/sec)                                |
| W          | weight (lbf)                                               |

### Subscripts

|           |                                              |
|-----------|----------------------------------------------|
| 0,1,2,3,4 | propulsion system burns & phases of flight   |
| f         | final                                        |
| i         | initial                                      |
| p         | propellant                                   |
| pay       | useful payload                               |
| s         | structure (including tankage & power supply) |

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